Apollo Thermal-Protection System Development

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The development of the Apollo thermal-protection system is presented chronologically. The paper discusses the avenues explored, the problems encountered, and the rationale which resulted in a blunt vehicle protected by a low-density ablator in a honeycomb matrix bonded to a steel sandwich structure. Extensive ground and flight tests and analytical studies were performed to support this development. A flight test program was designed to provide data at extreme conditions within the flight corridor, and the results of these flights were used to verify system performance and to refine prediction capability. It is concluded that structural and manufacturing considerations often outweigh thermal-performance considerations in system design, that many important-appearing problems are not significant, and that the technology gained in the Apollo Program forms a reliable base for the development of future thermal-protection systems.

Introduction

STUDIES for a manned lunar mission were initiated early in 1960 by the Space Task Group (STG) of NASA. At that time, Project Mercury was just beginning, and only boilerplate capsules had been flown. Thermal protection for return from orbit was considered feasible using heat shields of heavily glass-reinforced phenolic, but John Glenn's orbital flight was still 2 years away.

Early studies had shown that a simple ballistic vehicle returning from the moon to the atmosphere of the Earth had an entry corridor only 7 miles wide. A greater corridor width was considered necessary; therefore, a lifting vehicle would be needed. The lift capability would also give a ranging capability of several thousand miles and permit landing at a selected site. "Overshoot" trajectories ranging to about 5000 naut miles were therefore studied, along with a g-limited "undershoot" established at $20g$ and based on reasonable physiological expectations for crew survival (Fig. 1).

It was recognized that a relatively blunt vehicle would be required to minimize convective heating, and various blunted cones, lenticular shapes, and Mercury derivative shapes were explored. It was determined that a modest lift-to-drag ratio (L/D) (in the range of 0.3 to 0.5) would meet corridor and range requirements, but there was concern over the means of modulating lift. Volume requirements favored a long conical afterbody and considerations of afterbody heating favored a short cone well shielded by the forebody. A cone half-angle of 33° was considered a good compromise, and wind-tunnel tests showed that a vehicle with such an afterbody configuration would generate an L/D of 0.5 when at a 33° angle of attack and would demonstrate acceptable stability characteristics.

Most aspects of the thermal environment were reasonably well established by then-existing techniques, as described by Erb et al.[1] However, one unknown of concern was nonequilibrium radiation from the shock layer. Some estimates made during this time period (1960 and 1961) indicated that the nonequilibrium radiation might be orders of magnitude higher than equilibrium radiation and might be a dominant factor in entry heating. Further theoretical study and data from shock tubes and light-gas guns showed, however, that the importance of nonequilibrium radiation in comparison with equilibrium radiation decreased with increasing body size. Eventually, basing the Apollo Command Module (CM) configuration upon minimum convective heating was shown to be proper.

Early wind-tunnel data on the heating distribution verified that much of the CM experienced heating well below stagnation-point levels. Theoretically, it would have been possible to use the reradiative approach (e.g., the metallic shingles on Mercury) for a substantial area of the conical region. Mercury flight experience, however, had shown the difficulty of predicting heating in afterbody areas where there could be flow reattachment. The uncertainty about the level of heating over part of the body and the indeterminate problems of an ablator-radiator juncture at other than a surface discontinuity, such as at the Mercury shoulder, led to the choice of an all-ablative system.

Many thermal-protection materials were undergoing tests during the preliminary design phase in 1960 and 1961. The Mercury heat-shield material, a high-density (110 lb/ft^3) glass phenolic, was considered but dismissed because its high conductivity was ill-suited to a long entry. The NASA Langley Research Center (LRC), among other organizations, was testing nylon-reinforced phenolic in the 75-lb/ft^3 density range. This material, when described in the terms of "effective heat of ablation," looked promising based on thermal-performance tests in low-enthalpy arc facilities. The earliest estimate indicated that, at the stagnation point, 1.8 in. of phenolic nylon would be required. This would correspond roughly to a thermal-protection system (TPS) weight of 2900 lb.

During the winter and spring of 1960–1961, feasibility studies of the lunar-return mission were conducted by several aerospace contractors. Several materials were examined, in-

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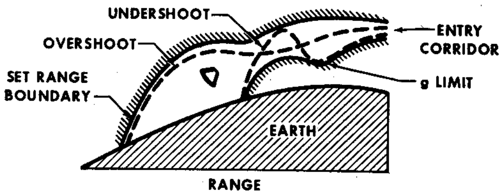

Fig. 1 Entry corridor.

cluding phenolic nylon, the General Electric Company "Century Series" of epoxy ablators, and the Avco Corporation Avcoat 5026-22 epoxy-novolac, a silica-fiber-reinforced material of 66-lb/ft^3 density. Avco also studied refractory metal honeycombs filled with various oxides which had the postulated advantage of high reflectivity to incoming radiation.

By the end of 1961, the CM contract had been let to North American Rockwell Corporation (NR) who had proposed to use phenolic nylon tiles over a supporting honeycomb sandwich structure of stainless steel. Various approaches were proposed in a subcontractor competition for the ablative tiles. Among the materials suggested were a phenolic-glass system similar to the Mercury heat shield, phenolic nylon, phenolic melamine, the "Thermo-lag" materials developed by Emerson Electric, the Century Series material, and the Avcoat 5026-22. Bonding of tiles, spray application, and fabric lay-up techniques were the approaches to manufacturing proposed by the various bidders. The Avco system was chosen.

Early Development

The TPS as initiated for the CM was an all-ablative system consisting of molded solid tiles of Avcoat 5026-22, approximately 1 ft^2, bonded to a shell of stainless steel honeycomb sandwich. This attachment method was later augmented with mechanical fasteners. The weight of the TPS was initially predicted to be 1680 lb.

The steel substructure was not the prime load-bearing structure, but simply reacted airloads to a cabin structure of aluminum. Differential expansions between the two shells were accommodated by a slip-stringer strain-isolation system (Fig. 2). The critical design conditions for the supporting structure were water impact for the blunt face and escape-rocket plume impingement during tumbling abort for the conical region.

The main design requirements established for the ablator were that it limit temperature at the ablator-steel interface (bondline) to 600°F during entry, that it be compatible with the steel substructure, and that it survive thermal cycling in the range from −250° to +250°F prior to entry. In addition, it was required to provide boost thermal protection and to withstand micrometeoroid, vacuum, and ultraviolet exposure.

A difficult problem arose because of the substantial difference in the coefficients of expansion of the ablator and the steel. The mating, bonding, and curing process gave a zero-strain condition at a temperature around 200°F. When cold-soaked to −250°F, the curved ablator panels tended to flatten. Curvature was maintained by the restraint of the cabin; however, this restraint caused tensile cracks normal to the surface in the ablator. Adjustment of the bonding technique could be made to accommodate the −250°F cold soak, but not while maintaining bond integrity at 600°F.

Meanwhile, efforts to reduce TPS weight were undertaken and lower density versions of the ablator were explored. Most promising was a 37-lb/ft^3 version (designated Avcoat 5026-39), and estimates of the TPS weight using this material were substantially lower than for the earlier material. Also, post-flight examinations of recovered Mercury heat shields, which had a 15-in.-diam center plug which was bonded in place (as would be an ablative tile), disclosed extensive areas where no bonding of ablator to substructure had occurred. The doubts thus raised regarding the ability to attach tiles led to a major review of fabrication and inspection techniques.

The Gemini heat shield, using a filled honeycomb, had been designed by this time and showed promise; therefore, a honeycomb approach was pursued for the Apollo Program. The Avcoat 5026-39 was applied by tamping it in mastic form into an open honeycomb of phenolic glass after the latter had been bonded to the steel and the bond inspected.

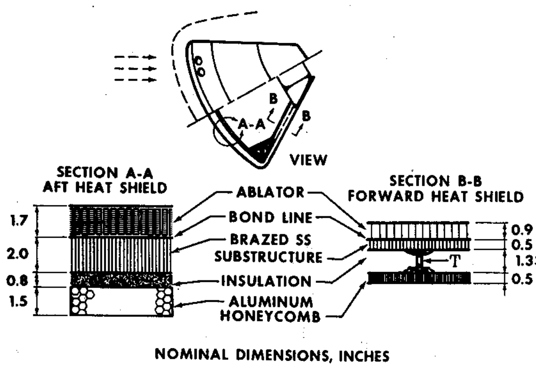

Fig. 2 Structural arrangement of the thermal-protection system.

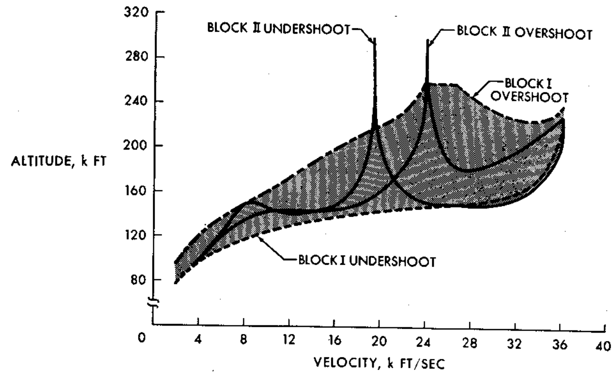

Fig. 3 Design entry trajectories.

Design and Manufacture

Thermal Design

Design trajectories were developed by NR, based on an L/D of 0.5 for the vehicle flying at a trim angle of attack of 33°, a lunar-return speed of 36,333 fps, and an entry along a flight path inclined 90° to the equator. A set of trajectories was defined, and two (Fig. 3) were termed the Block I design

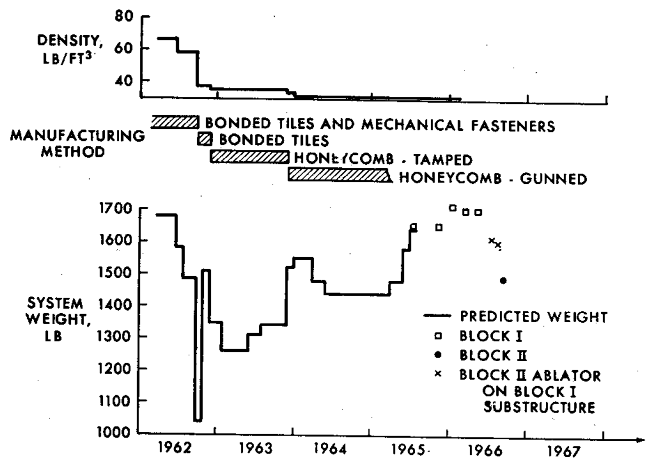

Fig. 4 History of ablator density and thermal-protection system weight.

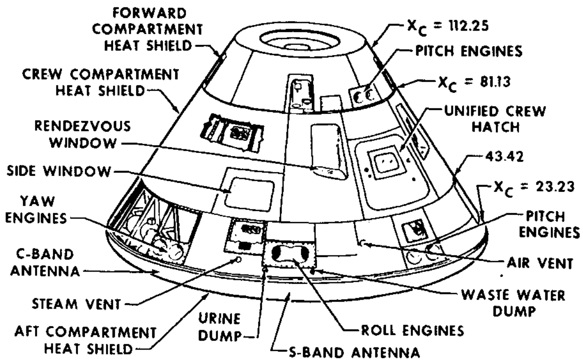

Fig. 5 Block II command module.

trajectories. (Block II trajectories are discussed later). The overshoot boundary was limited to a 5000-naut mile range, and an entry path was flown which would maximize convective heating. The undershoot boundary was predicated on a 20-g acceleration limit. Certain Earth-orbital entries were also examined, but the lunar-return overshoot governed TPS sizing over most of the spacecraft. This use of the most severe, aerodynamically feasible trajectories was considered a reasonable and not unduly conservative approach for the initial phase of the program.

The heating environment was defined at various body locations, and local heating effects due to surface irregularities were also defined. The thickness of the ablator was calculated at each body point by a subliming ablator program using a recession correlation based on the effective heat of ablation coupled to a solid-state conduction analysis.

As the design progressed, various refinements were made in the definition of heating and the extent to which the structure and various special regions were considered. These changes and the modifications to the manufacturing technique led to changes in ablator weight, as shown in Fig. 4. Generally, the increases were due to more severe general or local heating, to the effects of protuberances, and to unavoidable increments in thickness to achieve smooth fairings. Decreases were usually due to improvements in analysis or manufacturing and, as will be discussed, changes to the basic requirements.

Block II Modifications

In the fall of 1963, some refinements in CM design were undertaken which provided an opportunity to resize the TPS. A guidance technique had been established which utilized a

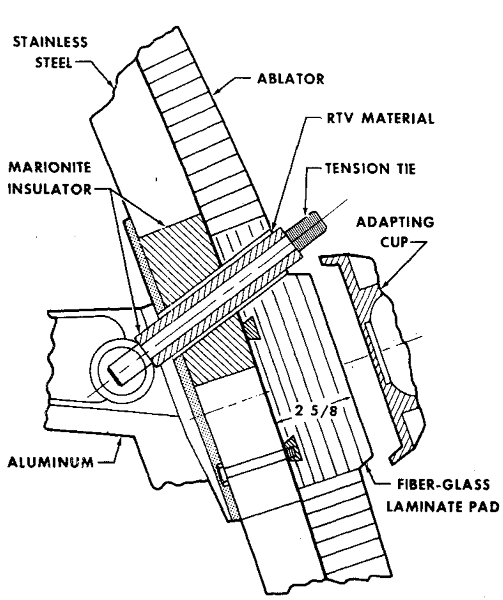

Fig. 6 Sectional view of shear pad.

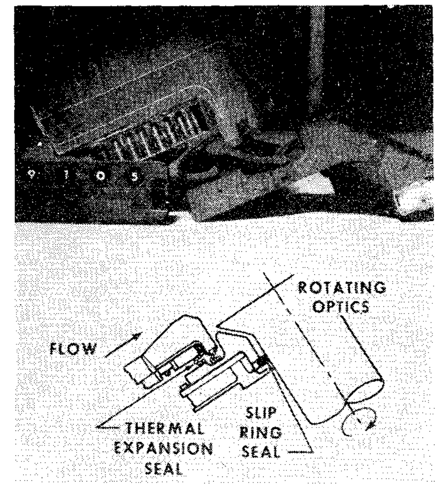

Fig. 7 Sectional view of astrosextant port.

skipout to acquire the desired range. In addition, it was decided to reduce the maneuvering range as a means of reducing TPS weight. The revised ranging requirement for design was 3500 naut miles instead of 5000 naut miles, which included a 1000-naut miles downrange maneuvering capability, primarily for weather avoidance. The design had also progressed to a point where packaging the CM to obtain a center-of-gravity offset sufficient for a 33° angle of attack was no longer possible. An L/D range of 0.3 to 0.4 was chosen. The design value for return inclination to the equator was also reduced from 90° to 40°. The lower value was considered a more realistic operational limit since landing in cold regions was to be avoided. The new set of design trajectories is mapped in Fig. 3.

By this time, it had been established that cracking of the ablator would occur at temperatures lower than about $-170°$ F. It was possible to relieve the $-250°$F limit since, for a variety of reasons, a mode of passive spacecraft temperature control had been adopted by which control over severe temperature excursions was maintained by a slow rolling "barbeque" maneuver to distribute solar energy over the spacecraft.

Concern over the visibility through the CM windows, following exposure to the exhaust of the escape-tower rocket, led to the implementation of a cover to protect the windows during boost. It appeared little more complicated to cover the entire conical section and to make it possible to remove approximately 0.2 in. of ablator which had been provided for boost thermal protection. Also, with a boost cover, a temperature control coating could be applied to limit the upper temperature to $+150°$F and to further reduce the severity of design conditions for the TPS and for the environmental control system.

The Block II CM also incorporated a docking mechanism which necessitated a flat apex. The configuration which resulted is shown in Fig. 5.

Special Regions

The many heat-shield penetrations, which were required for structural attachments, guidance and navigation (G&N) optics, extravehicular activities (EVA), and other functions, represented unique design problems. The particular treatment accorded to these singular regions can be illustrated by the CM/service-module (Fig. 6) attachment, which consists of six pads of high-density fiber glass surrounded by low-density ablator with steel tie bolts extending through three of the pads. The bolts penetrate a reinforcing block, a densified region of the steel honeycomb structure, and a fiber glass tension-tie plate below the pad assembly. Surrounding the imbedded portion of the tie bolt is an insulating sleeve of Marinite.

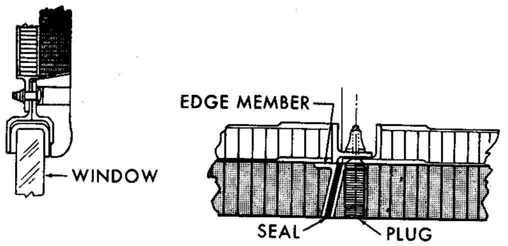

Fig. 8 Manufacturing approach in special regions.

During the Block II design period, the heating at protuberances was studied in detail[2] and it was concluded that wholly rational treatment of protruding pads was not possible. A design resulted in which all special elements of the TPS were recessed or mounted flush with the ablator surface. In the case of the shear/compression pads, the pad was initially thinner than the ablator (Fig. 6). The adapting cup seated in an annular groove, and at no time during entry did the pad become a protuberance. The other special regions were treated in similar fashion, and flush or recessed antennas, umbilicals, et cetera were incorporated. Only the EVA handrails were permitted to protrude on the Block II vehicle.

Relatively late in the program, certain changes in the CM design necessitated new approaches to parts of the TPS. In 1966, the use of an articulating door over the G&N optics was abandoned in favor of a passive system. The passive astro-sextant and telescope installation, located on the windward conic, had to allow a component attached to the cabin interior to extend through the ablative heat shield, remain free to rotate, and maintain some degree of movement along the axis of rotation. The unique features of the design which resulted are the thermal seals. One type of seal facilitates rotation while preventing penetration of the thermal environment by the use of a small slip ring. A flexible-type thermal seal prevents the flow of hot gas into the cavity between the unprotected cabin and the TPS wall (Fig. 7) while allowing movement of the TPS with respect to the cabin.

Manufacturing

The current manufacturing scheme evolved rapidly after the shift to filled honeycomb. Early items were manufactured by tamping the ablator into the cells, but a device resembling a grease gun was later developed to inject the material into the cells, with each cell being individually filled. The layup of the honeycomb core is accomplished by fitting and splicing prior to bonding, and all raw edges (such as at windows) are closed out with molded fiber glass detail parts (Fig. 8). Because of the heterogeneous nature of the ablator, some voids often resulted. The assembly is therefore x-rayed and faulty cells are refilled prior to cure. The empty honeycomb is machined to final contour prior to filling with ablator (Fig. 9). After filling, the surface is sanded to remove any excess ablator.

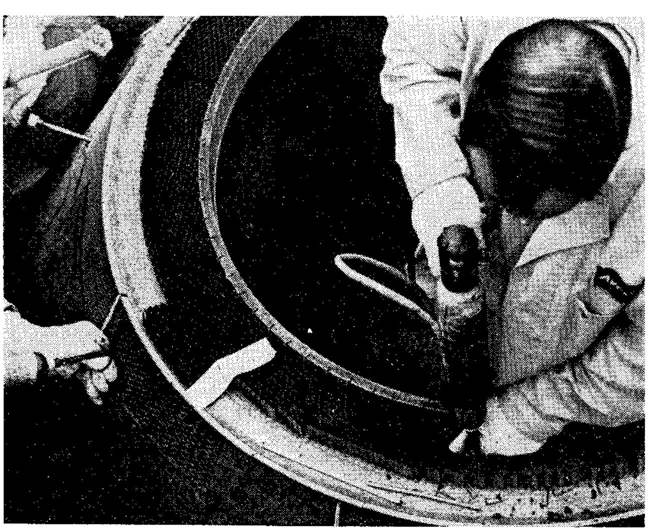

Fig. 9 Gunning ablator.

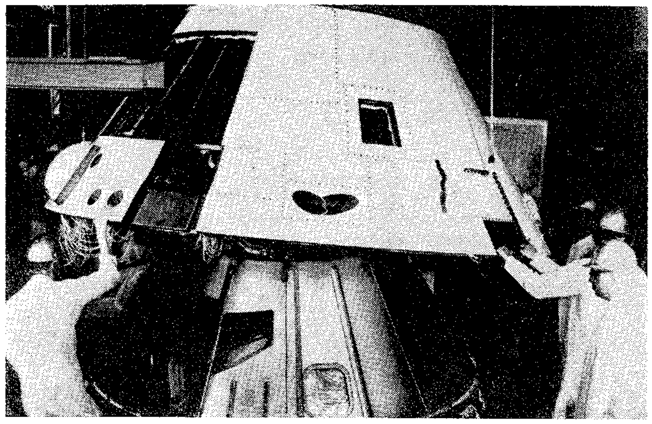

Fig. 10 Crew compartment thermal-protection system.

Cured-state repairs are sometimes required for voids which are not identified in the green state. More detail on the manufacturing is included in other publications.[3] The cured assembly is then sealed against moisture and painted (Block I) or supplied with an adhesive-backed temperature control coating (Block II). Gaskets made of room-temperature vulcanizing elastomer are provided around doors and access panels and are poured and cured in place. Figure 10 shows the finished crew compartment of the TPS being fitted to the CM cabin.

Ground Testing and Analysis

The basis for ablator thickness design rests on predictive models of thermal performance which, in turn, hinge on data from ground tests in a simulated entry environment and conventional thermal properties. Avco performed extensive tests of the ablator in various facilities[3,4] and measured surface and char interface recession and temperatures in depth for a variety of heating rates enthalpies, pressures, and shears. The results were applied by analytical representations involving a thermochemical "heat of ablation" expressed as a function of environmental energy and a surface temperature expressed as a function of heating rate. The heat-of-ablation approach was considered fast and convenient for design, but a more complex analysis was considered necessary to verify the design. The more elaborate approach included the complete energy balance at the surface and an accurate treatment of the char characteristics.[4,5]

Numerous tests were conducted to determine strength and stiffness properties over a wide range of conditions and, generally, to express them as functions of temperature.[6] Next, tests were run on panels and components ranging in size up to a few feet square. Temperature cycling was carried out on panels with ablator applied to both sides of the honeycomb sandwich to simulate complete restraint in bending. Vibration and structural-strength tests were also performed on such panels to obtain the behavior of the composite. Component-level testing was also done on panels which included important singularities, for example, window and escape-tower-leg recesses. Typical "specification" tests for salt spray, fungus, etc., were also carried out.

Several of the CM test articles included either the steel substructure of the TPS or the complete system. A structural test article without ablator was used in static tests to demonstrate the capability to withstand design loads and temperatures. Water and land drop tests were conducted at varying conditions, again with and without ablator.

The TPS was also tested under thermal-vacuum conditions. Both Block I and Block II command modules underwent simulated missions, each several days long, in the Space Environment Simulation Laboratory at the NASA Manned Spacecraft Center. These tests exposed the spacecraft (under vacuum conditions of approximately 10^{-6} torr) to the temperature extremes expected in flight. Heat was provided by carbon-arc solar simulators, and a liquid-nitrogen cold wall provided the appropriate heat sink. These tests[7] established the extent to which gaps between compartments opened, the extent to which the shell distorted, and the extent to which the strain-isolation system was exercised as various temperature and temperature-gradient conditions were applied.

On the Block I CM, some cracking of the ablator was experienced at a temperature warmer than the minimum design level. However, subsequent ablation tests of cracked ablator panels showed that cracking per se was not a problem. On the other hand, it was recognized that delamination could constitute a problem. Coupon pull tests were conducted at the crack locations and no evidence of delamination was found. The observed cracks were attributed to a prestress condition arising from the manufacture and assembly. Cracking at the warmer temperatures was not observed during a subsequent thermal vacuum test of a Block II spacecraft under similar conditions. The conclusion from these tests was that the thermostructural performances of the TPS is acceptable.

Flight Testing

The basic uncertainty which would exist if only ground tests were available is on the order of ±40% on thickness of the ablation material. Suitably instrumented flight tests conducted as the program progressed were necessary to permit refinement of the analytical performance prediction. Three flight tests were planned to qualify the TPS. The first two were entries at slightly greater than orbital velocity and demonstrated the performance of the TPS for the nominal operational conditions of manned orbital flights. They were conducted on Block I command modules and used the uprated Saturn I launch vehicle which gave a capability for entry at relative velocities up to about 27,000 fps. The entry trajectory for the first flight, designated AS-201, was chosen to provide the highest possible heat-transfer rates, and hence the highest ablator-surface temperature and surface-recession rates that could be achieved with the uprated Saturn I vehicle. For the second flight, mission AS-202, the entry trajectory was designed to provide the highest possible heat load and to provide the two-pulse entry typical of a lunar return. The entry trajectories of both test spacecraft were very close to the planned trajectories. Data on the entry conditions are given in Table 1.

To demonstrate the performance of the TPS for nominal lunar return, a test was conducted using the Saturn V launch vehicle which provides a capability for full lunar-return entry speeds. This flight, designated AS-501, was conducted with a CM which had a Block I structure but had the Block II ablator distribution. It was determined that a nominal mission could be flown which would give a high heating rate comparable to the lunar mission undershoot trajectory.

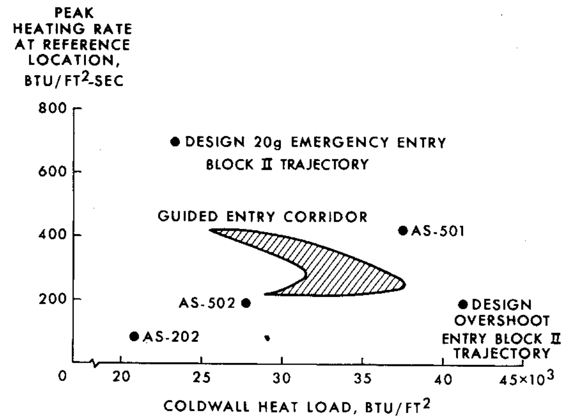

Fig. 11 Entry conditions for Apollo command module.

The launch-vehicle development program required two unmanned flights of the Saturn V, thus one further test of the TPS at lunar-return speed was planned. This entry, mission AS-502, achieved a relative velocity of only 31,530 fps because of a launch-vehicle malfunction. It did, however, provide data at entry conditions between those of Apollo missions AS-202 and AS-501 (Fig. 11).

Ablator Performance

The TPS was instrumented with calorimeters and with thermocouples located at varying depths in the ablator. Postflight measurements were made of surface recession, depth of char penetration, and the density variation with depth. Data from the flight instrumentation were continuous through touchdown for all but the first flight (AS-201), on which an instrumentation system failure occurred 50 sec into the entry. Flight data are reported in other publications.[8–11]

Following each flight, adjustments were made to the analytical model[12] to reconcile its results to the data. The results of missions AS-201 and AS-202 provided a basis for adjusting abaltor properties, notably conductivity. However, since little change of ablator thickness occurred, no information could be derived about the surface recession characteristics. Data from missions AS-501 and AS-502 defined a char densification effect due to deposition of essentially pure carbon near the surface. When this effect was included in the analysis, good agreement with the flight data resulted.[12] The conclusions from the flights to data is that some conservatism exists in the TPS design as a result of compounded conservatisms in heating, ablator performance properties, and manufacturing-tolerance assessments. However, the actual margin is not large in comparison with, say, structural margins.

Table 1 Mission summary

Mission	Entry velocity (relative), fps	Entry velocity (inertial), fps	Entry angle (inertial), deg	L/D	Range, naut mile	Entry time, sec	Reference q, Btu/ft²-sec	Reference q, Btu/ft²
AS-201	25,318	26,482	−8.60	0.332	470	674	164	6,889
AS-202	27,200	28,512	−3.53	0.275	2295	1234	83	20,862
AS-501	35,220	36,545	−6.945	0.365	1951	1060	425	37,522
AS-502	31,530	32,830	−5.85	0.350	1935	1140	197	27,824
AS-205	24,497	25,846	−2.072	0.305	1584	935	59	11,281
AS-503	35,000	36,221	−6.48	0.300	1350	868	296	26,140
AS-504	24,603	25,896	−1.76	0.310	1835	1004	53	13,725
AS-505	34,968	36,314	−6.54	0.305	1295	871	296	25,728

Behavior of Singularities

The special regions of the CM were instrumented only sparsely, and the main results from flight test are qualitative observations on their appearance. All such regions have been found to be thermally compatible with surrounding areas of ablator. Where internal temperatures were measured, the response was modest and the effects of the singularities on adjacent regions of ablator were negligible. A few instances of specific performance might be noted. On mission AS-502, the side hatch on the leeward conic suffered only discoloration of the paint. The recessed umbilical on the windward conic displayed only discoloration and no in-depth heating effects. As expected, all the tension-tie bolts melted nearly flush with the shear-pad surfaces. No significant charring of the ablator around the bolts was exhibited nor was there any significant temperature response at the base of the bolt.

The general performance at singular regions also displays some conservatism. However, while it is clearly possible to design safe methods of treating penetrations such as on the Apollo spacecraft, the detailed understanding of the performance and heating in such regions is limited.

Thermostructural Response

A serious attempt was made to exercise the strain-isolation system during mission AS-501. The CM conic region was provided with thermal control coatings and the spacecraft was oriented to the sun so that the windward side would soak cold and the leeward side would soak hot. Thus, on entry, maximum thermal distortion of the thermal-protection shell with respect to the cabin would occur.

No instrumentation was carried to measure stress levels or shell displacements, but the overall performance indicated that the strain-isolation system performed satisfactorily under the most severe conditions.

Monitoring and Supporting Programs

North American Rockwell Corporation Backup Program

Late in 1962, concern grew over the weight of the TPS. The honeycomb approach was firmly established by then, but it was hoped that some other filler might offer a weight advantage. Accordingly, a backup program was initiated at NR. Extensive tests for thermal performance and for thermostructural properties were undertaken for five materials, including the Avcoat 5026-39. The other materials were the Gemini material d.c. 325, a silicone with Eccospheres; a General Electric elastomer, ESM-1000; Emerson Electric Thermo-Lag T-500-13; and "purple blend," developed by NASA LRC, a silicone with phenolic microballoons in addition to Eccospheres. The results of the testing showed that thermal performance was much alike for all materials with a variation about the mean of TPS weight of only $\pm 15\%$. As only very simple analytical models were used to correlate results and to predict system weights, no clear thermal advantage was shown for any material.

Thermostructural tests for thermal cycling were also conducted on flat panels of the various materials. It quickly became apparent that the existing low-temperature requirement of $-250°F$ could not be tolerated by some ablators. Only the Avcoat 5026-39 could withstand the cold-soak condition at all thicknesses and, accordingly, it was confirmed as the mainstream TPS material.[13]

NASA Manned Spacecraft Center Backup Programs

Early in 1962, an alternative approach for the conical region TPS was initiated with Chance-Vought. This approach involved a pyrolized graphite or carbon-cloth laminate (called burnt toast) as the exterior element and a honeycomb core filled with very low-density (20 lb/ft^3) ablator as the insulating element. As ablation occurred, gas escaped through the porous exterior element. The system was developed and ground qualified on panels approximately 2 ft^2 (Ref. 14). By the time this effort had been carried to completion in mid-1963, the mainstream program was proceeding satisfactorily and the burnt-toast approach was dropped, although it could have been advantageous if integrated from the start with the structural design.

A second backup development was pursued with spray-applied versions of Thermo-Lag, designated T500-4 and T500-6. These versions differed from each other only in that the T500-6 version was lightly reinforced with a loose-weave cloth. Thermal and structural properties were obtained and generally promising performance was shown.[15]

By the end of 1963, a renewed concern over TPS weight and new data on possible adverse thermal performance of the Avcoat 5026-39 in a shear environment forced another look at alternative approaches. The NR backup program had explored the T500-13 material, a heavy version (89 lb/ft^3) which had poor thermostructural performance. The generally good thermal performance demonstrated by the spray-applied version led Emerson Electric to develop a lighter (35 lb/ft^3) version which could be filled into a honeycomb and which could withstand cold soak well. The lighter version, designated T500-111, was the basis for a major backup program conducted by Emerson Electric for NASA MSC in the early part of 1964. The result was a well-characterized material fully adapted to the Apollo manufacturing scheme and compatible with a steel substructure in the cold environment.[16,17] No very worthwhile thermal-performance advantage had been shown, however, and a flight of the Avcoat 5026-39 material on the Scout vehicle[18] had eliminated the concern over shear sensitivity, at least for the time being. Accordingly, this backup program was terminated.

Monitoring Testing

The state of the art in simulating entry environments was such in 1962 as to commit NR and NASA MSC to broad parallel efforts in thermal-performance testing. Whereas Avco did some 5000 tests during the program, NR and NASA MSC each conducted perhaps 20% as many.[19–23] In retrospect, this gave all groups much greater confidence but would hardly be considered necessary for a program started now. Some testing was done to determine effects of micrometeoroid impact and the degradation of thermal performance due to penetrations.[24,25] Evaluations were made of the hazard caused by micrometeoroids, and the general conclusion was that probable impact damage represented no significant hazard to the TPS.

Supporting Flight Program

There is no way within the present-state-of-the-art to exercise the TPS fully prior to flight. It is comforting, therefore, to obtain some sort of flight verification prior to committing a full-scale test as costly as an Apollo mission. Two flight-verification programs were conducted on small-scale vehicles.

A Scout vehicle was used to test the ablator at lunar-return heating rates. However, to match the heating rates at less than lunar-return velocity resulted in a more severe pressure environment. Some ground tests at the higher pressures showed very poor performance for the ablator, and by flight time, there was great concern on the part of some for the survival of the payload.

The Scout flight, made in the summer of 1964, was successful and demonstrated, within the Apollo range of pressures, the general validity of the design data and approach. This conclusion, however, was almost overshadowed by the higher than predicted recession rates experienced at high pressure near the end of the flight. Further, the intricate analysis required to demonstrate the adequacy of the material for Apollo

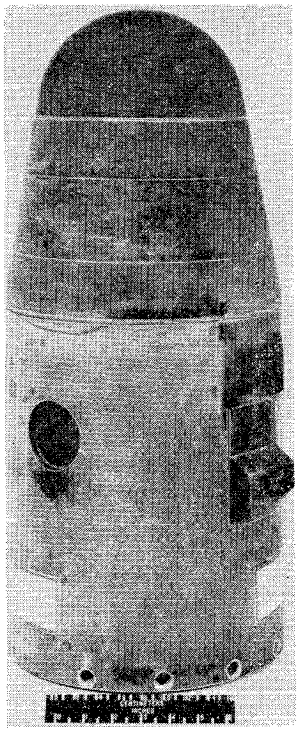

Fig. 12 Pacemaker.

environments took much effort. It can be concluded that if a developmental flight test is undertaken at conditions significantly different from the design conditions, careful interpretation must be given to negative results in order to avoid extraneous issues and unnecessary anxiety.

A special flight test was made to evaluate the design adequacy of various protuberances and singular regions and to obtain ablator performance data in a turbulent, high-shear stress environment. This test was carried out on a Pacemaker vehicle (Fig. 12). Although it followed the first two tests of the Apollo spacecraft itself, it was at rather more severe conditions.[26] Thus, it contributed to confidence in the design.[27–30]

Supporting Analysis

Following the same monitoring philosophy for analysis as for test, both NASA MSC and NR[31–33] developed analytical models of the charring ablation materials used for the TPS. It is typical of such models that, while ground-based test data are used for their construction, it is necessary to validate them by so adjusting critical parameters (typically char conductivity) that predictions and flight results can be reconciled.[34] Additional analyses were done at NASA Ames[35] and NASA LRC.[36,37]

While these multiple efforts produced some spirited controversy, the net result was greatly increased understanding of ablation phenomena. The technology of ablation analysis was much advanced during this period, and a sound capability for predicting how the Apollo material would behave was developed.

Current Status

The TPS is wholly satisfactory and both NASA MSC and NR have confidence that this complex system will continue to perform well during the Apollo Program. The Apollo 8–13 lunar missions have been successfully concluded since the original presentation of this paper. However, the guidance and propulsion systems performed with faultless accuracy on these missions, and the entries of these vehicles were near optimum rather than near design conditions. Considering the critical nature of the system, the investment in extensive testing, backup programs and replication of analysis is considered to have been well worthwhile. With the clarity of hindsight, however, it is believed that more effort should have been concentrated on mainstream program support and somewhat less on parallel and backup programs.

Conclusions

A TPS for the Apollo CM has been designed, built, and flown on 11 missions. The system performed well on all flights, five of which involved a lunar return. The experience from Apollo 8 and Apollo 10–13 has shown that an operational choice is made in favor of very short range entries and that the CM can be guided very precisely to midcorridor flight-path angles. This results in an entry environment much less severe than that for which the TPS was designed. The major conclusions derived from this experience are as follows.

1) After various screening and backup programs, it is apparent that a) current ablators of the same general density range (35 to 55 lb/ft^3) have comparable thermal performance, b) simple analyses do not form a sound basis for assessing TPS weights, and c) flight testing in environments not closely representative of design conditions can cloud issues and cause unwarranted concern if not carefully interpreted.

2) Thermal performance of the ablation material is one of the lesser criteria in developing a TPS. Viewing the system as a whole, major changes were made for inspection access, thermal stress, manufacturing, L/D, and performance at singularities but never to obtain better thermal performance of the basic ablator.

3) An adequate technology exists to allow the design of ablative thermal-protection systems for manned spacecraft entering at speeds at least as high as those associated with lunar return. However, uncertainties in technology made necessary conservative assumptions which yielded undefined margins of safety. Significant advances have been made which now allow the rational design of near-optimum ablative protection. However, special effort is needed to understand heating and TPS performance in singular regions.

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